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ON THE SIMULATION OF TRANSONIC SHOCK-TURBULENT BOUNDARY LAYER I--ETC(U)
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ON THE SIMULATION OF TRANSONIC SHOCK-TURBULENT BOUNDARY LAYER INTERACTIONS IN CRYOGENIC OR HEAVY GAS WIND TUNNELS†

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ON THE SIMULATION OF TRANSONIC SHOCK-TURBULENT BOUNDARY LAYER INTERACTIONS IN CRYOGENIC OR HEAVY GAS WIND TUNNELS†

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Abstract

The role of the basic similitude parameters governing transonic normal shock-turbulent boundary layer interaction effects in cryogenic wind tunnel tests is studied theoretically for the non-separating case. Besides Mach and Reynolds number, these parameters are the wall to total temperature ratio, specific heat ratio γ, viscosity-temperature exponent and Prandtl number. The results show that lack of temperature ratio simulation has a significantly adverse effect on interactive skin friction and hence separation onset compared to the adiabatic free flight case; higher γ's than air also may have some effect.

† Based on work supported by the Office of Naval Research under Contract N00014-75-C-0456.

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NOMENCLATURE

C_f	skin friction coefficient $(2\tau_w/\ _{oe}U_{oe}{}^2)$
C_p	pressure coefficient $[2(p-p_o)/\rho_{oe}U_{oe}{}^2]$
H	boundary layer profile shape factor (δ^*/θ^*)
M_1	$= M_{oe}$ = Mach number of incoming inviscid flow
P	static pressure
Pr	Prandtl number
Re_L	Reynolds number based on U_{oe} and L
r	recovery factor
T	absolute static temperature
u,v	velocity components in x and y directions, respectively
x,y	coordinates parallel and normal to surface
β	$(M_1^2 - 1)^{1/2}$
γ	ratio of specific heats
δ	boundary layer thickness
δ^*	boundary layer displacement thickness
ΔP	static pressure jump across incident shock
η	non-dimensional displacement surface perturbation $(\Delta y/\delta_o)$
μ	coefficient of viscosity
ω	viscosity-temperature exponent $(\mu \sim T^{\omega})$
ρ	density
θ^*	boundary layer momentum thickness

<u>Superscripts</u>

$(\ )'$	disturbance quantity (with respect to incoming flow)

Subscripts

e freestream properties upsteam of shock at boundary layer edge

o denotes undisturbed (not stagnation) flow property ahead of shock

L length of undisturbed shock location

w property at wall value

i i = 1, 2, 3, denotes region of variable (Fig. 1)

INTRODUCTION

It is well known that the aerodynamics of missiles, entry vehicles and aircraft flying at transonic speeds is significantly influenced by the interaction between local normal shock waves and the viscous boundary layer flow. Consequently, it is important to insure that this flow feature is adequately simulated in experiments carried out in future cryogenic or heavy gas high Reynolds number wind tunnels, especially since their cost per data point is expected to be higher than usual.[1] In addition to the flight Mach and Reynolds numbers which are by design simulated in such facilities,[1,2] four other parameters are involved in the interaction effect on a transonic flow field which may not be duplicated owing to the very low temperature - high pressure working fluid involved: wall to total temperature ratio T_w/T_t, specific heat ratio γ, viscosity temperature dependence exponent $\omega(\mu \sim T^{\omega})$ and the Prandtl number Pr. Of these, the first is deemed especially important since the aerodynamic tests in some proposed short duration cryogenic transonic wind tunnels result in the model being at much higher temperature than the flow total temperature during much of the test. Moreover, the specific heat ratio of cryogenic nitrogen flow can be significantly higher ($\gamma \sim 1.5$-1.8) than air,[3] as can that of argon ($\gamma = 1.67$), and this may have some influence on the interaction effects since γ is involved in both the shock and boundary layer governing equations. Specific heat ratios lower than air ($\gamma \sim 1.1$) are also of interest in facilities[4] using heavy gases such as Freon 12.

Although the influence of real gas effects on the inviscid transonic flow[3,5] and on laminar boundary layer-supersonic oblique shock interaction[5] has been examined in detail, an appraisal of the more important turbulent-normal shock interaction problem has yet to be made. Existing theoretical studies of transonic shock turbulent boundary layer interaction to date have been confined to the case of adiabatic flow in normal air. To appraise the influence of the aforementioned similitude parameters, the present paper therefore describes the extension and application of a recently-developed basic analytical theory of the nonseparating transonic interaction of a weak normal shock with a turbulent boundary layer to include the effect of heat transfer and arbitrary values of γ, ω and Pr. Results are given for pressure distributions, flow geometry and skin friction.

OUTLINE OF THE THEORETICAL MODEL

For unseparated turbulent interacting boundary layers ($M_1 \lesssim 1.3$) it is possible to construct a fundamentally-based theory of the problem. In its original form the flow is taken to consist of a known adiabatic zero pressure gradient boundary layer profile $M_0(y)$ subject to small transonic disturbances due to an impinging weak normal shock. The theoretical model of this interaction emerges from an asymptotic analysis of the compressible Navier-Stokes equations at High Reynolds numbers, giving a linearized inviscid boundary value problem surrounding the nonlinear shock discontinuity and underlaid by a thin viscous disturbance sublayer as schematically illustrated in Figure 1. As described in detail elsewhere,[6] an approximate analytical solution can be achieved in the leading approximation by introducing suitable simplifying assumptions which are physically sound provided $M_1 \equiv M_{oe}$ is not too close to unity ($M_1 \gtrsim 1.05$); solutions can then be obtained by operational methods for each subregion indicated in Fig. 1. An arbitrary value of γ may be retained in so doing. Detailed comparisons[6,7] have shown that the results give a good account of the important engineering aspects of the interaction (e.g., $\partial p/\partial y$ effects across the boundary layer near the shock, interaction pressure rise along the wall, displacement thickness growth and interactive skin friction) over a wide range of Mach-Reynolds number conditions at very low computational cost. It is also readily adapted to non-adiabatic flow conditions with arbitrary values of ω and Pr, and hence is well-suited as a basis for the present study.

The matching of the aforementioned regional solutions yields readily-solved linear integral equations for the disturbance pressure along both the boundary layer edge and wall.[6] The remaining interactive flow properties can then be determined in terms of p'_w. For example, double integration of the combined continuity and y-momentum equations across the width of the inviscid rotational disturbance region in the boundary layer yields the local increase in displacement thickness. The viscous disturbance-flow solution in the underlying viscous sublayer determines the interactive skin friction perturbation associated with p'_w. Thus, upon correcting for the important non-linear inertia effects in an adverse pressure gradient using the general non-dimensional wall shear-pressure solution ahead of separation given by the triple deck theory[8] (converted to turbulent flow by expressing all results in terms of C_{f_o} instead of Re_L), it is found that

$$C_f(x) \simeq C_{f_o} - \sqrt{\beta C_{f_o}\left[1 + \left(\frac{C_f(x) - C_{f_o}}{1.234 C_{f_o}}\right)\right]}\; C'_{p_w}\; F(x/\delta_o) \tag{1}$$

where the non-dimensional function F is essentially unity ahead of the shock $x < 0$ and vanishes slowly behind it with $F \sim (x/\delta_o)^{-1/3}$ far downstream. It is seen from Eq. (1) that depending on Reynolds number (C_{f_o}), a sufficiently strong interactive pressure rise can cause incipient separation ($C_f \to 0$) near the shock.

It can be shown[9] that the presence of small-to moderate heat transfer in transonic flow does not significantly influence either the inviscid or viscous disturbance fields by introducing any new perturbation equation terms; in the leading approximation it enters only implicitly as it influences the undisturbed $M_o(y)$, skin friction C_{f_o} and boundary layer thickness δ_o. The boundary layer profile used is based on an accurate flat plate turbulent eddy viscosity model with arbitrary ω that is well-suited for interaction studies[10]; it uses a modified Crocco integral to compute the temperature and hence provides for an arbitrary degree of surface heat transfer with a recovery factor $r \approx P_r^{1/3}$ and arbitrary γ.

It is noted that the purpose of this study is to delineate the overall engineering parameter sensitivies; accordingly, we have deliberately used a very simplified real gas model involving constant equivalent thermodynamic (γ) and transport (ω, Pr) properties. Should their effect be significant, it is understood that a more detailed treatment of the thermodynamic and transport relations within the interaction flow analysis might be necessary.

DISCUSSION OF RESULTS

To provide a general basis for appraising the similitude requirements of various facilities with different gases and/or thermal histories (ranging from fan-driven or blow-down to short duration Ludwieg tubes), a wide range of the parameters was studied. The following are typical results.

Heat Transfer Effect

The predicted wall temperature effect on the interaction pressure distributions along the wall is illustrated in Fig. 2 and is seen to be quite weak; as expected, this was found to be true over a range of shock strengths and Reynolds numbers. Increasing wall temperature tends to slightly increase the upstream influence and lower the pressure immediately downstream of the shock with a tendency to lengthen the asymptotic "tail". Regarding the important property of the characteristic upstream influence distance x_{up} (here defined as the distance upstream of the shock where the local interaction-induced pressure rise is only 5% of the overall total), we have found[9] that over a wide range of conditions the ratio $x_{up}/\delta_0 \approx O(1)$ and is only slightly affected by heat transfer; thus, e.g., cooling reduces the upstream influence essentially proportional to the corresponding reduction in δ_0, as also observed at higher supersonic Mach numbers.[11]

The interaction-induced growth of the boundary layer displacement thickness is of great practical interest since this often has a significant back-effect on the inviscid flow and shock position on an airfoil or in channel flows.[7] Typical results are shown in Fig. 3 illustrating the expected thinning out with increasing Reynolds number or wall cooling. The influence of a hot wall $T_w > T_{w,ad}$ is increasingly significant at lower Re_L. Note also that this effect is significant in terms of the ratio $\Delta Y/\delta_0$; since δ_0 is itself increased by heating, it is even larger in terms of ΔY.

The effect of shock-boundary interaction on the local skin friction is of particular importance in transonic airfoil design and testing, since it bears directly on the downstream boundary layer behavior and its possible separation. Since wall temperature influences the undisturbed skin friction C_{f_0} (see Table 1), the relative effect on its interactive decrement alone can be shown by plotting the ratio $C_f(x)/C_{f_0}$ in Fig. 4A. Owing to the interaction-induced adverse pressure gradient, C_f/C_{f_0} typically decreases downstream toward the shock with a minimum occurring slightly behind it*, followed by a subsequent

* Although no longer valid for separated flow where $C_f(x) < 0$ over some portion of the wall, the present theory is still useful to indicate trends toward this situation, i.e., where and when incipient separation ($C_f \to 0$ at some x) first occurs.[7,9]

gradual rise further downstream. It is seen that wall temperatures appreciably different from adiabatic significantly influence the interactive-reduction of C_f and hence the onset of incipient separation; in particular, hot walls are predicted to magnify the adverse effect of the interaction of C_f and hasten the occurance of incipient separation under the shock, whereas wall cooling has the opposite beneficial effect. Judging by comparison with calculations showing the effect of Reynolds number (shown in Fig. 4B), it would appear that proper wall temperature simulation may be of comparable importance to Reynolds number as regards skin friction.

The foregoing results are in qualitative agreement with experimental data on non-adiabatic interactions at supersonic speeds with oblique shocks.[11] However, to the authors knowledge, there exists as yet no experimental data on transonic unseparated interactions with heat transfer.

Specific Heat Ratio

Although studies have shown that the purely inviscid aspects of the transonic flow field are not significantly affected by the value of γ in the range 1.1 - 1.8 (Ref. 3-5), coupling with viscous effects as occurs in shock-boundary layer interaction may change this conclusion and hence warrants investigation.

The results of our calculations show that there is only a barely-discernable effect of γ on the interactive wall pressure distribution over a wide range of Mach and Reynolds numbers: as shown in Fig. 5, increasing γ slightly expands the nondimensional streamwise scale of the interaction. Likewise, there is only a very small change in the interactive-thickening of the boundary layer (Fig. 6). These predicted insignificant effects are in agreement with experimental observations[4] and similar to theoretical results for the laminar oblique shock interaction case.[5]

The influence of γ on the interactive skin friction, however, is more interesting: Fig. 7 shows the typical effect on the ratio $C_f(x)/C_{f_o}$. Increasing γ reduces not only the undistrubed C_{f_o} but also even more the interactive decrement so as to moderately hasten the onset of incipient separation. The explanation for this evidently lies in the noticeable increase of the adiabatic wall temperature with γ (see Table 1); increasing γ thus has an adverse effect on C_f qualitatively equivalent to that of a higher wall temperature. Combined with the foregoing wall temperature effect, this lack of air property simulation in a cryogenic facility with significantly higher γ may be important where skin friction and incipient separation effects are significant, especially in view of the drastic change in the basic interaction flow pattern when separation occurs[13-16] (Fig. 8).

Viscosity Exponent and Prandtl Number

Examination of available thermophysical property data for cryogenic nitrogen[12] indicated that over the working temperature range of aerodynamic testing interest, a power law viscosity-temperature relationship $\mu \sim T^{\omega}$ is a reasonable approximation, with $\omega \approx .75$ and decreasing slightly with decreasing absolute temperature. To assess the sensitivity of the viscous interaction to ω, calculations were made for $\omega = .5$ (classical kinetic theory), .75 and .80 ; the results showed a completely negligable effect on all the interaction properties of physical interest including the skin friction.

Likewise, Prandtl number data[12] shows little change from the usual air value ($\approx .72$) over a wide temperature range down to nearly the liquifaction limit; since the influence of Pr is felt through the basic turbulent boundary layer recovery factor ($r \approx Pr^{1/3}$), negligable influence of this parameter on the interactive similitude would be expected. Calculations in which r was varied from .8 to 1.0 verified this.

Upstream Boundary Layer History

In addition to their direct effect on the interaction, the aforementioned similitude parameters also may have a significant indirect effect through the influence on the upstream boundary layer history as reflected in the incoming boundary layer profile shape. For example, it is wellknown[13] that shock-boundary layer interaction can be very sensitive to the laminar-turbulent transition history including tripping devices and surface roughness (Fig. 9), and these in turn may be influenced by wall heating and specific heat ratio effects.[17] The role of the pressure history ahead of the stock may also be very significant: Fig. 10 illustrates this with some results obtained by Panaras and Inger[18] that show the appreciable change in normal shock-turbulent boundary layer interaction properties associated with small variations of the initial shape factor. Thus, if the upstream pressure gradient effect on the boundary layer (especially if adverse) is influenced by T_w and γ, as the shape factor values shown in Table 1 suggests it could be , this must be included in the similude appraisal of the subsequent interaction region.

CONCLUDING REMARKS

The present study has used a basic theoretical model as a tool to appraise the influence of nonadiabatic wall temperatures and lack of specific heat ratio simulation on the shock-boundary layer interaction aspect of the transonic flight regime in either flight or wind tunnel applications. For example, the sensitivity of skin friction predictions to $T_w/T_{w,ad}$ and γ suggests that the lack of freeflight adiabatic wall temperature ratio and gas property simulation in a cryogenic tunnel may significantly exaggerate the interaction effect and its attendant flow separation on the model compared to a flight case at the same M_1 and Re_L. Consequently, some basic experimental studies of transonic interactions under non-adiabatic conditions appear desireable to test these conclusions.

It is noteworthy that, aside from broadening our basic understanding of viscous-inviscid interations, there are two other practical applications where the results of the present study may be of interest: (a) post-entry transonic flight phase of the Space Shuttle orbiter, where transonic shock-boundary layer interactions can take place on the hot surface caused by the entry heating history; (b) studies of transonic flow around cooled turbine blades operating in hot gas flow conditions.

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TABLE 1

Wall Temperature and Specific Heat Ratio
Effects on Undisturbed Boundary Layer Properties

$M_1 = 1.20$, $Re_L = 10^6$ $(\omega = .76, Pr = .72)$

T_w/T_{e_o}	$C_{f_o} \times 10^3$			
	$\gamma = 1.10$	$\gamma = 1.40$	$\gamma = 1.67$	$\gamma = 1.80$
.50		4.04		
.75		3.71		
1.00		3.44		
ADIABATIC (T_w/T_e)	3.47 (1.07)	3.23 (1.26)	3.10 (1.39)	2.98 (1.52)
1.50		3.08		
2.00		2.81		

T_w/T_{e_o}	$(\delta_o/L) \times 10^2$			
	$\gamma = 1.10$	1.40	1.67	1.80
.50		2.51		
.75		2.37		
1.00		2.27		
ADIABATIC	2.26	2.20	2.17	2.15
1.50		2.15		
2.00		2.08		

T_w/T_{e_o}	FORM FACTOR H			
	$\gamma = 1.10$	1.40	1.67	$\gamma = 1.80$
.50		1.07		
.75		1.42		
1.00		1.76		
ADIABATIC	1.64	2.19	2.44	2.76
1.50		2.40		
2.00		3.09		

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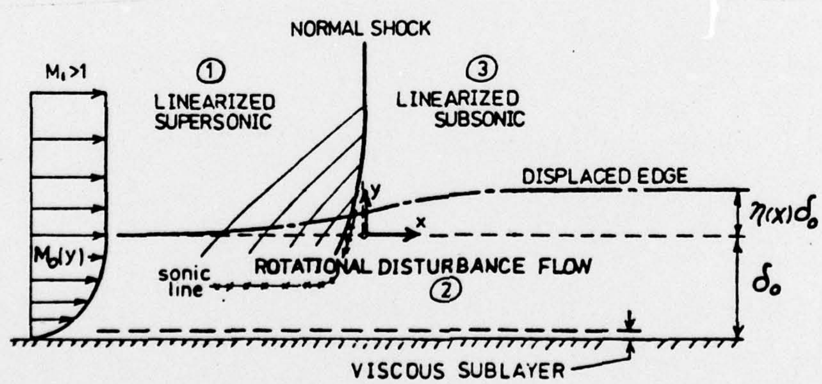


Fig. 1

Normal Shock-Turbulent Boundary Layer Interaction Flow Model (Schematic)

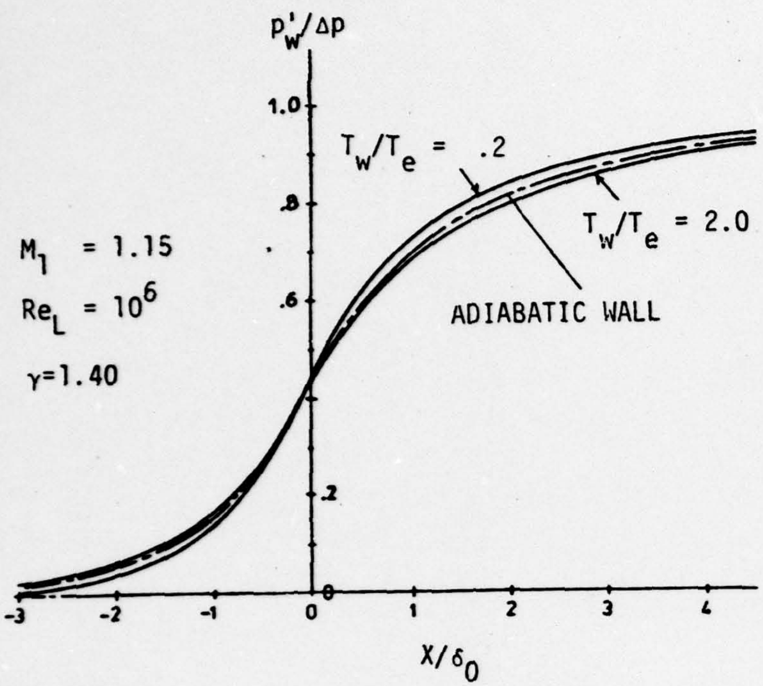


Fig. 2

Wall Temperature Effect on Surface Pressure Distribution

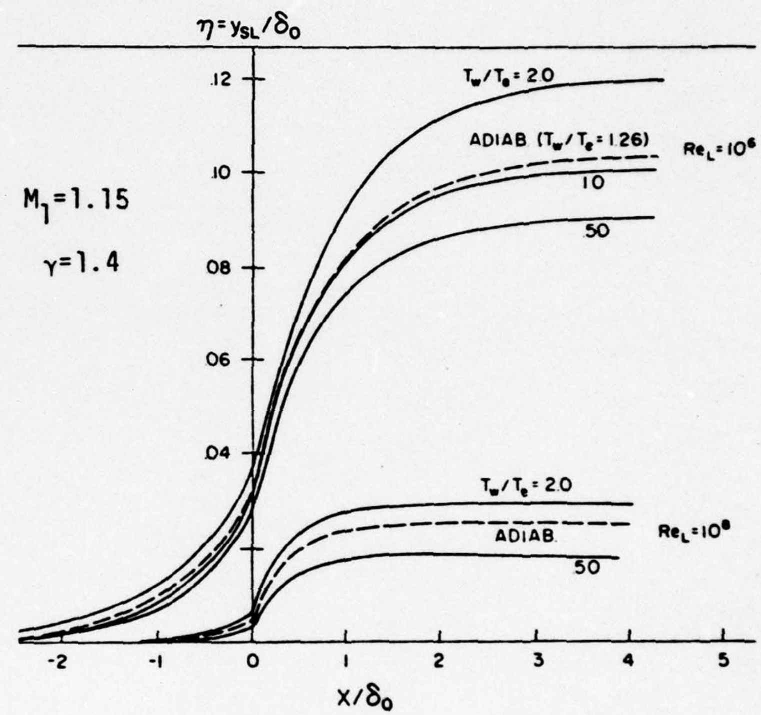
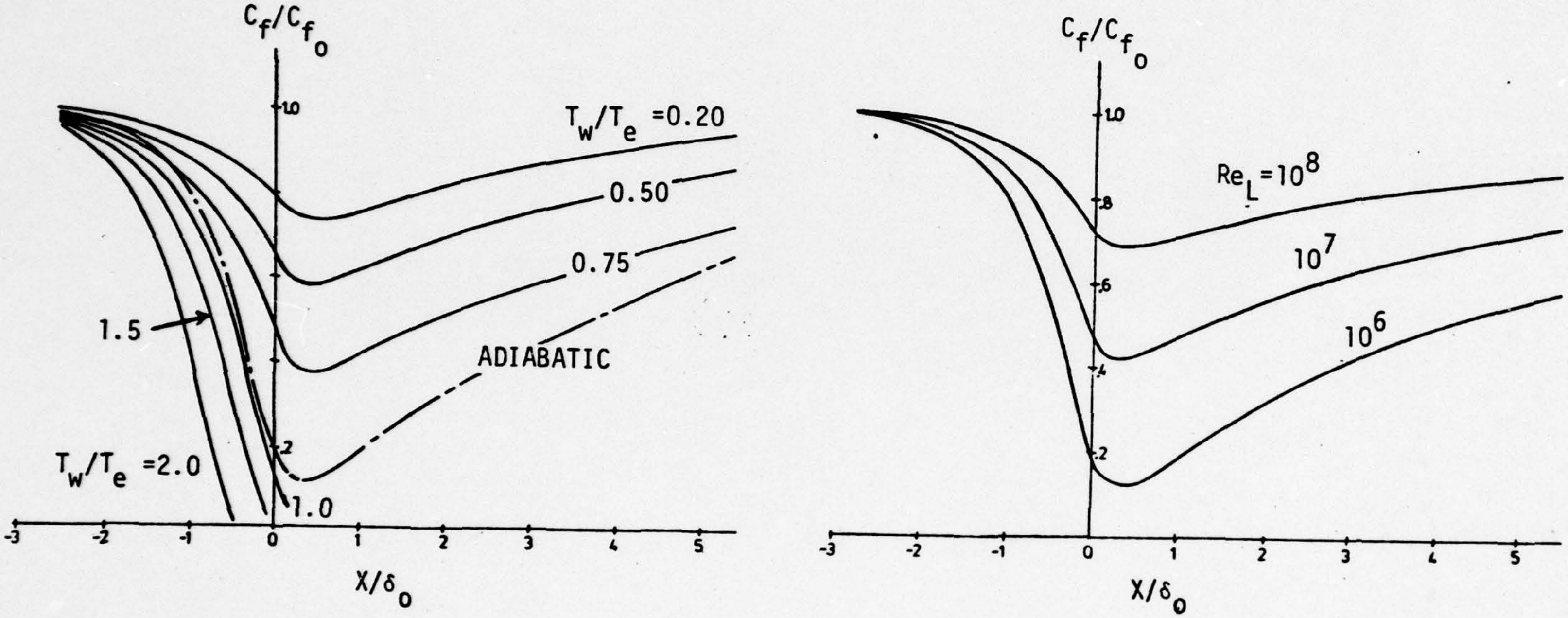


Fig. 3

Wall Temperature Effect on Interactive Thickening of the Boundary Layer



A. Wall Temperature Effect

B. Reynolds Number Effect

Fig. 4. Interactive Skin Friction Distributions

(M_1=1.15, $Re_L=10^6$, $\gamma = 1.4$)

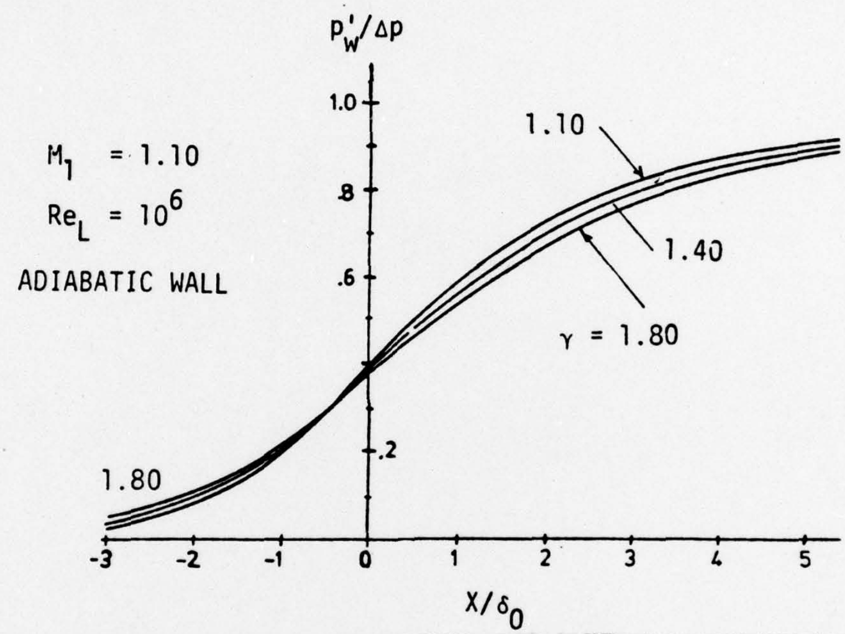


Fig. 5

Specific Heat Ratio Influence on Wall Pressure

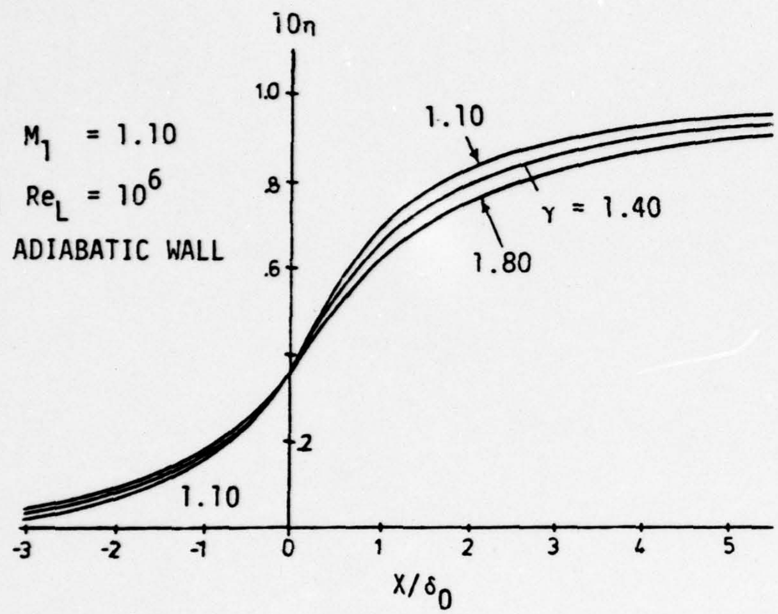


Fig. 6

Specific Heat Ratio Effect on Interactive Thickening

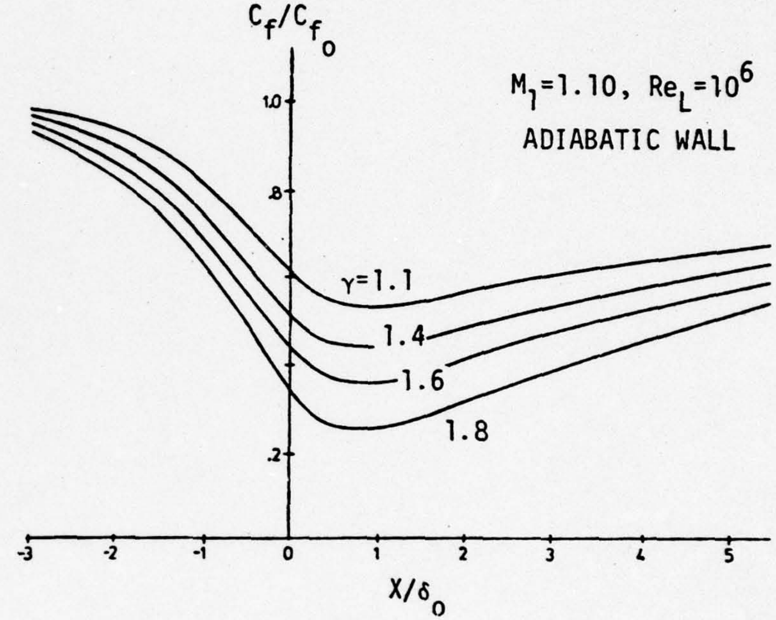


Fig. 7

Specific Heat Ratio Effect on Interactive Skin Friction Distributions

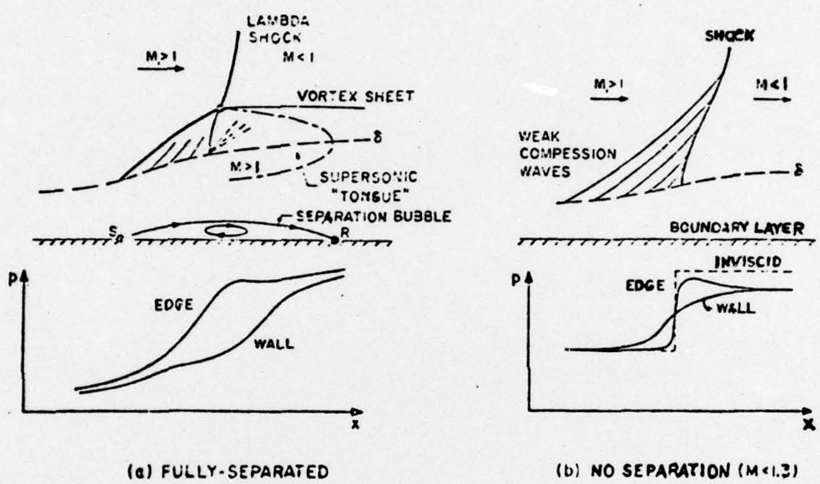


Fig. 8

Influence of Separation on
Interaction Flow Pattern

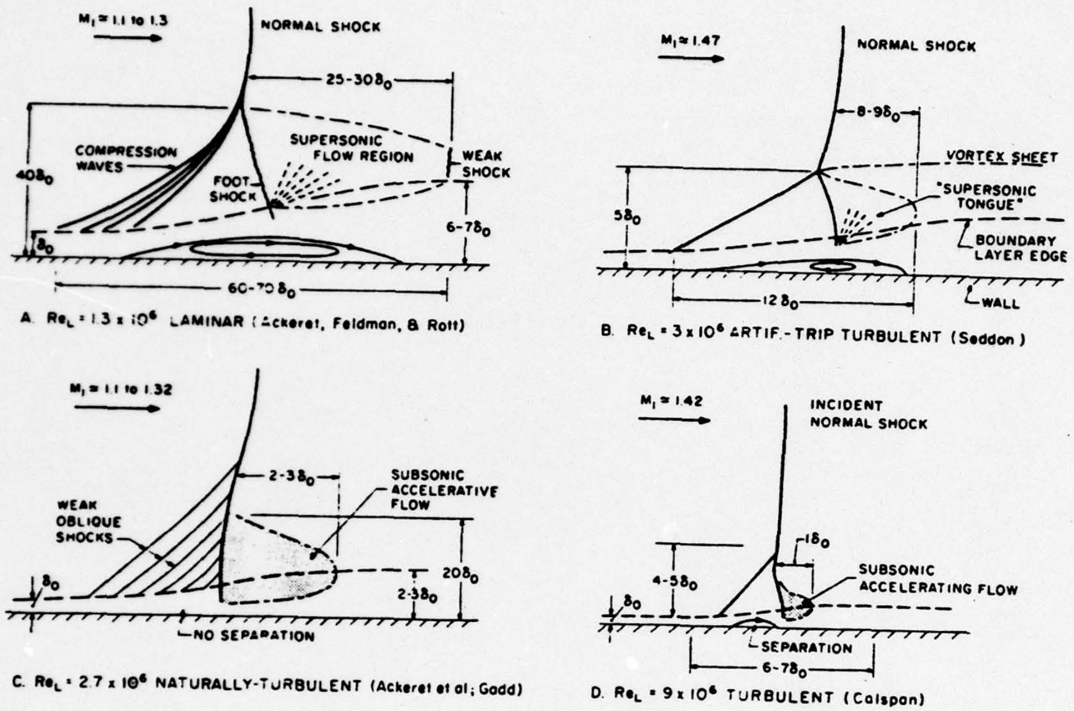


Fig. 9

Upstream Laminar-Turbulent Transition Effect on the Shock-Boundary Layer Interaction

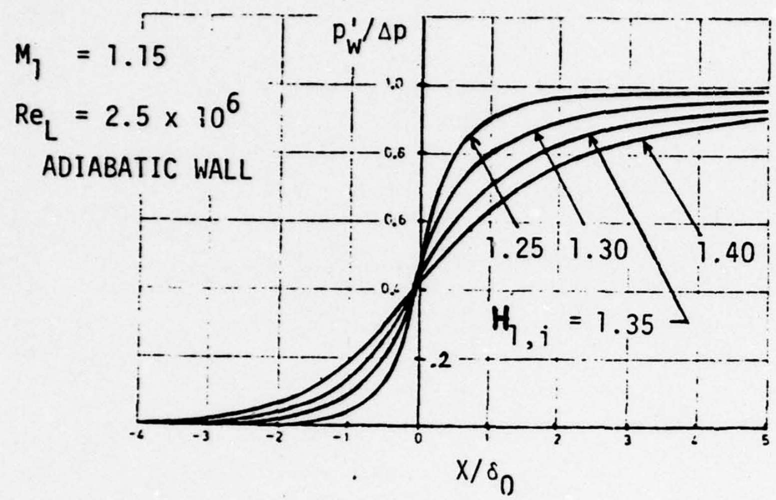


Fig. 10

Typical Shape Factor Effect

on Interaction Pressure

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The role of the basic similitude parameters governing transonic normal shock-turbulent boundary layer interaction effects in cryogenic wind tunnel tests is studied theoretically for the non-separating case. Besides Mach and Reynolds number, these parameters are the wall to total temperatures ratio, specific heat ratio γ, viscosity-temperature exponent and Prandtl number. The results show that lack of temperature ratio simulation has a significantly adverse effect on interactive skin friction and hence separation onset → over

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